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SCIENCE AND RECONSTRUCTION

One of the greatest dangers in our conception of the present-day world is that we may be so concerned with problems of temporary shortage that we fail to see the fundamental importance of the long-term plan.

A proper balance must be struck between the consumption and the capitalisation of our scarce resources and we must avoid being either too wasteful, on the one hand, or too miserly on the other.

At the present time there is a grave maladjustment, not only in the relationship between demand and supply, but also in our basic economic position, for insofar as we are living on a fast diminishing (and *borrowed*) capital, the full effects of the position have not yet appeared. But when they do so our difficulties will undoubtedly be multiplied a hundredfold by such factors as the growth in average age of the population, reduction in the number of working hours, a growing loss of industrial efficiency, and vast schemes of a social nature to which we have recently committed ourselves.

After a conflict which has demanded so much from every one of us, there is a natural tendency to look for better conditions and a natural resentment in finding a situation of shortages, petty disputes and general uncertainty which overshadows industry.

To resolve from this actual and even greater potential chaos a new order of world beauty demands planning of the most imaginative kind. Science is faced with a challenge more inspiring and calling for greater ideals than it has ever faced before. Vast problems are rising on every side, in industry, in economics, in society, and in every branch of human life.

But we have compensatory advantages. To combat these problems we now possess a vast army of young men, each enriched with a greater knowledge of human affairs, of conditions in other parts of the world, and very many with a wealth of scientific experience and a capacity of improvisation which is truly remarkable. Many of the old customs have gone by the board and perhaps many more are yet to go.

Into our new plan must be woven the things we have striven for, the schemes of town planning, mechanisation of industry, the use of new materials and processes, a greater equality and a greater opportunity for all, improved medical and educational services, and a new conception of scientific thought which is to supply its motive power in every branch of human welfare.

For too long has science been forced to develop from seeds of war. It is the duty of our new generation of scientists to ensure that science is allowed to develop also in times of peace as the great benefactor that she really is.

PROBLEMS IN ROCKET DEVELOPMENT

By J. Humphries, B.Sc., G.I.Mech.E.

A paper read to the British Interplanetary Society in London on December 7, 1946

Introduction

My intention in this paper is to give an indication of the main problems facing the rocket designer and show how these problems have been solved in various rocket units, and also to give an indication, where possible, of other methods of solution. Before starting on the main part of the paper I will give an idea of where rocket propulsion fits with respect to other types of propulsion. Because of the low propulsive efficiency at low speeds the applications to which rockets are most adapted are aircraft and projectile propulsion. Even for aircraft the field is comparatively restricted and the chief uses are for take-off and for high subsonic and supersonic flight. Projectiles come almost exclusively in the high subsonic and supersonic flight region and thus the rocket seems to have a much wider application to this type of work than to aircraft.

Jet propulsion devices as a whole can be divided into two main classes, those units utilising atmospheric air, and rockets, whose operation is independent of outside air. Rocket units can be further broken down into three types, gaseous, liquid and solid propellant types. Gaseous units are not suitable for flight installation, due chiefly to the low density of the fuels, but have been used to a fairly large extent for experimental purposes for studies of rocket motor characteristics. Liquid fuel units are extensively used both for aircraft and projectile propulsion and will be considered here as offering the greatest interest to engineers. Solid fuel units have been used widely for projectile and aircraft assisted take-off work, but are confined almost entirely to short burning single firing units.

The rocket propellant consumption is the highest of all power plants. At the present time it is of the order of 17 lb./hr./lb. thrust compared with a fuel consumption of 1·1 lb./hr./lb. for a turbojet unit. However, there are a number of applications in which the rocket unit competes favourably with other power plants, either because of its high impulse/weight ratio, or because it can supply a high thrust for a short period combined with a maximum of simplicity. The chief fields of use of liquid propellant rockets are:—

(1) assisted take-off and performance boosting of aircraft powered by other types of power plant;
(2) as sole propulsion plant of high speed aircraft;
(3) propulsion of projectiles;
(4) propulsion in empty space—here the rocket is the only suitable device and the high altitude sounding rocket is now being used in the U.S.A. for cosmic-ray investigations up to heights of over 100 miles.

Figure 1 shows the basic component parts of a liquid propellant rocket unit Two systems of fuel supply are shown, both being enclosed by a broken line The main components of the unit are:—

(1) The motor—this converts the chemical energy of the propellant into

heat energy and then transforms the heat energy into kinetic energy of flow of the exhaust gases. The motor itself is made up of a propellant injector, a combustion chamber and an exhaust nozzle.

(2) Propellant control valve. This is used as a means of starting and stopping the propellant flow to the motor. A variable thrust can be obtained by varying the flow through the valve.

(3) Propellant feed system. The system consists of tanks for propellant storage and a pressurising system. Pressurisation can be obtained by means of power-driven pumps or by a regulated gas pressure from tanks or a gas producer.

I shall deal chiefly with the choice of propellants, the rocket motor, and the propellant feed system where most of the problems peculiar to rockets occur.

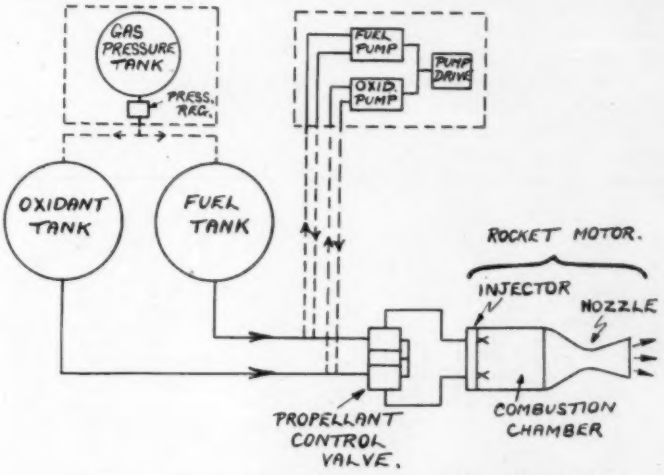

FIG. 1. Component parts of liquid propellant rocket unit.

Fuels

The first step in designing a rocket unit, having determined its general characteristics, such as its application, thrust and time of running, is to decide on the propellants to use. Here we are faced with a vast range from which to choose—over 3,000 different propellant combinations have been tested by the German concern of B.M.W. alone—not all successfully. However, there are certain desirable characteristics to be sought after, and before examining specific combinations it will be as well to look briefly at some of these characteristics.

The most important requirement is a low consumption, or to use a term more commonly employed in rocketry, a high "specific impulse"; the specific impulse being the thrust obtained from the consumption of one unit of propellant mixture per second. Next come ignition lag, rate and ease of reaction,

stability of the propellants, density, vapour pressure, specific heats in the liquid phase, corrosivity, availability, toxicity, inflammability, freezing point, viscosity and cost. To deal with the most important characteristic first, Figure 2 shows a number of possible propellants together with their theoretical specific impulses, and density impulses. The density impulse is discussed later. The order of specific impulse obtained with present-day fuels is 200 seconds, which, from a glance at this table would appear to be very low compared with possible figures. The reasons for this low figure are elucidated later in this paper. The specific impulse for a given gas mixture for an adiabatic expansion is given by

$$\text{S.I.} = \sqrt{\frac{2\gamma}{g(\gamma-1)} \frac{GT_c}{M} \left[1 - \left(\frac{p_e}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

where $\gamma = \frac{c_p}{c_v}$.

G = universal gas constant, ft. lb./lb. mol./°C.
M = mean molecular weight of exhaust gases.
T_c = chamber temperature, °K.
p_c = chamber pressure.
p_e = exit pressure.

Propellants		SI_{th} *secs.*	SI_d *secs.*
Lithium	Fluorine	667	574
Lithium	Hydrogen	662	193
Beryllium	Oxygen	660	942
Beryllium	Fluorine	642	770
Boron	Oxygen	631	946
Lithium	Oxygen	617	478
Aluminium	Fluorine	587	802
Magnesium	Fluorine	583	753
Decaborane	Oxygen	579	712
Aluminium	Oxygen	554	905
Hydrogen	Oxygen	550	242
Magnesium	Oxygen	539	771
Silicon	Oxygen	525	778
C_8H_{18}(Petrol.)	Oxygen	434	477
Carbon	Oxygen	412	523
Ethanol	Oxygen	397	401
Black powder	—	210	346
"C-stoff"	"T-stoff"	203	243
"T-stoff"	—	107	143

FIG. 2. Properties of some propellant combinations.

The main variable in this expression is the factor T/M and Figure 3 shows the variation of the specific impulse with T/M for an expansion ratio of 20 for two

values of γ, namely 1·2 and 1·3. It is seen that the variation of γ has a comparatively small effect on the specific impulse whereas T/M has a great effect. Thus the most useful propellants from a practical point of view are those with end products having a low mean molecular weight, as this means that for a given specific impulse a lower chamber temperature is required for these fuels than for those giving gases with a higher mean molecular weight. Chief interest, therefore, centres on those fuels containing a fair percentage of hydrogen, particularly readily available hydrocarbons such as petrol, kerosene, diesel oil, and methyl and ethyl alcohols. As nozzle efficiencies of the order of 95 per cent. are now obtainable one would expect specific impulses of the order of 350–400

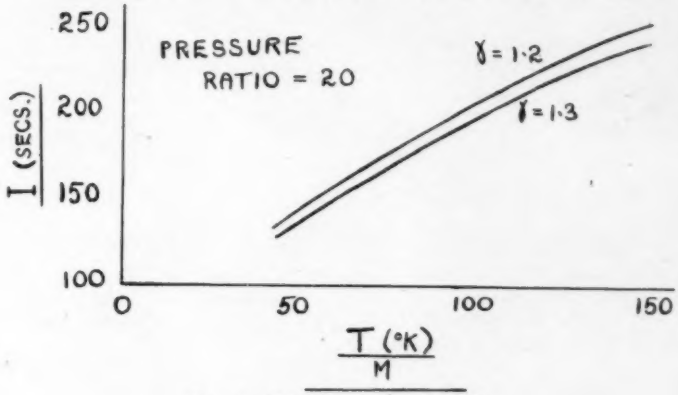

FIG. 3. Variation of I with T/M.

to be obtained with these fuels, but unfortunately before the temperatures associated with these figures are attained dissociation occurs, consequently allowing only about 60 per cent. of the latent energy to be released as heat energy. This dissociation can be reduced by increasing the combustion chamber pressure, but this results in an increase in weight of the unit as a whole, and an optimum value is reached above which any gain in specific impulse is offset by the corresponding increase in weight of the unit. At the moment, with available materials, we are limited, in the smaller sizes of rocket at any rate, to temperatures of the order of 2,000°C.–2,500°C. and so would need to develop materials and means of cooling before the temperatures associated with these high impulses could be tolerated. These temperatures obtainable from hydrocarbons are of the order of 3,500°C.

The boron hydrides form a group not so far investigated and lie between boron and hydrogen in much the same way as the hydrocarbons lie between carbon and hydrogen. They should consequently be far more effective than pure hydrogen when burnt with oxygen, whilst the other properties of the boron hydrides are not much different from those of the hydrocarbons. We especially know that many boron hydrides are liquid at normal temperatures and pressures and consequently can be pumped and handled easily. Even the

normally solid substances such as the light metals can be fed by pumps if they are mixed thoroughly with liquid fuels, for instance hydrocarbons. Hereby, at the same time, a difficulty is removed connected with the very high boiling points of most of the products of these special substances. The expansion of the burning gases, consisting, for example, of pure metal oxides, in the nozzle can continue only so long as these are gaseous. After the liquefaction takes place between 2,000 and 4,000°C. further expansion is impossible, and the heat content belonging to the lower temperatures is lost. If, however, the expansion takes place together with combustion gases of CO_2 and H_2O, the liquid or solid

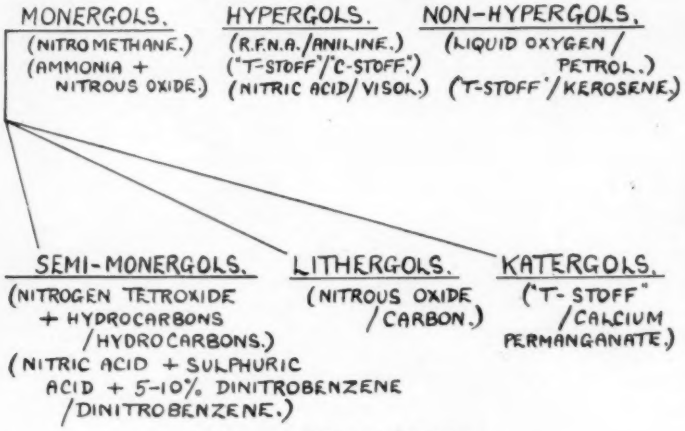

FIG. 4. Propellant classification.

particles can deliver their heat to adjacent molecules with lower boiling points, and by heating these, the lower temperature ranges of the metal oxides can be made useful. The boiling points of the fluorides mentioned in the list, without exception, lie lower than those of the metal oxides. It seems, therefore, that the fluorides could be used alone. However, for the metal oxides also the effective temperature difference between the combustion temperature and the boiling point is so great that they, even alone, will possibly give good specific impulses.

Coming back from the fairyland of specific impulses of 500 secs. it is of interest to examine some of our present day fuels in more detail and see some of their various advantages and some of the problems connected with them. Fuels are broadly classified in three groups, monergols, hypergols and non-hypergols (see Fig. 4). *Monergols* have the characteristic of containing both the oxygen carrier and the combustible in one fluid. A monergol may be a mixture or a single substance, but in any case it is stored in a single container and is injected without mixing any other component in the combustion chamber. An obvious advantage of the monergols is that they eliminate any need for mixing nozzles or jets, also the complexity of the pumping, controlling and feeding system is

cut by half compared with the common two component propellant mixture. The great disadvantage connected with them is that they are very sensitive to detonation and temperature, and up to the present this has been their greatest drawback. With this type of propellant the reaction has to be initiated by one of the methods described later under non-hypergols. A variation of this type where an attempt has been made to keep the good points of monergols while doing away with their bad stability is the *semi-monergol* group. Here the main propellant is of the monergol type but its oxygen content is not adjusted to the stoichiometric ratio, i.e. chemically correct ratio. Thus for the complete exploitation of the propellant it must be used with a second fluid which makes up the deficiency, either of oxygen or fuel as the case may be. By this means several advantages can be obtained, e.g. reduction of chamber size. In order to accelerate combustion it is always advantageous to ensure pre-mixing, as by this means the ignition delay and reaction time can be shortened; a shortening of the reaction time means that thermodynamic equilibrium is more quickly reached and thus a smaller combustion chamber is required. This pre-mixing and consequent reduction of chamber size is one of the advantages of semi-monergols. The danger of detonation and general instability of a monergol which makes its use as a coolant rather hazardous can be overcome, as certain mixtures of oxidants and fuels, normally useless as monergols because the mixture is far from the stoichiometric ratio, can be used. A further advantage of a semi-monergol is that the freezing point, boiling point, density, etc. can be varied, within a certain range, to suit one's requirements. An ethanol magnesium perchlorate semi-monergol fuel was developed for V.2. due to the shortage of methanol. This fuel had a higher density, higher specific impulse and was cheaper than any other fuel proposed for V.2. Semi-monergols, like monergols, require initiation of the reaction.

A *lithergol* process consists of the injection of a single fluid, the oxygen carrier, into the combustion chamber, the combustible being fixed in the chamber as a solid insertion. Oxygen carriers used in tests at B.M.W.'s were nitrous oxide (>90 per cent.) and hydrogen peroxide (<85 per cent.) Again as with monergols, there is only one fluid injected into the chamber so that there is no need for mixing jets or nozzles. Control is simple as far as this point is concerned. The solid combustible used in the first B.M.W. experiments consisted of hollow, cylindrical tubes of carbon fused or cemented together for insertion into the combustion chamber. One difficulty was the necessity for getting ignition of the combustible over its whole surface. Spotty ignition could lead to chamber explosion. A limitation on rockets of such a type is that they cannot be built for large impulses, as the walls of the combustion chamber get heavier as the impulse increases, until the total weight of the unit becomes prohibitive. Lithergol systems appear to be worth further investigation for short burning times, as for example in assisted take-off units. A further development in this direction was to do away entirely with the solid combustible insertion and to use a catalyst for the decomposition of the oxygen-carrier; such a propellant arrangement is called a *katergol*.

The distinctive property of *hypergols* is that the reaction is self-starting. The basic idea behind the development of this class was the realisation that

previous rocket experiments using non-self-igniting fuels had frequently led to chamber explosions, due to an accumulation of propellants in the chamber before ignition could be obtained. By having components which would react rapidly, without outside help, an excess of reactants in the chamber could be avoided. One of the earliest hypergols was hydrogen peroxide as oxygen carrier and hydrazine hydrate as combustible. This mixture has been very widely used in Germany and the well-known Walter 109–509A.2 used for propelling the Me.163B used a development of this mixture, namely 80 per cent. hydrogen peroxide, known as T-stoff, and a mixture of hydrazine hydrate, methanol and water together with a small amount of catalyst, known as C-stoff. However, it must be emphasised that even for hypergols there is a characteristic time delay for ignition to occur after mixing. This acceptable time delay should be less than 0·1 sec. at all conditions under which starting is required. There is no difficulty in obtaining propellants with much shorter time lags, especially at normal temperatures.

Finally there is the class of *non-hypergols*, or non-self-igniting propellants, the largest class of all and including all of the very early experimental propellants. As mentioned earlier one of the drawbacks of this class is the possibility of the propellant mixture accumulating in the chamber and so causing an explosion on ignition. Due to this same reason of not igniting on contact they have the advantage, at least for piloted aircraft, that in case of accident causing the fuels to come in contact with each other, they will not explode. A drawback of non-hypergols is that an equivalent process to "lighting the blue touchpaper" is required. Under certain circumstances this can be achieved quite easily by introducing a flame into the chamber from an external source. An example of this used on a large scale was the V.2. where a pyrotechnic device rather similar to a catherine wheel was ignited in the venturi just prior to turning on the propellants. For most applications though a built-in igniter is required and several methods for achieving ignition have been tried or suggested. The first method is by a built-in electrically fired cordite charge; this is suitable only for one-shot firing. Secondly, a spark or glow plug projecting directly into the chamber can be used although this also is very often only suitable for one start due to the high chamber temperature burning away the projecting part of the plug. This trouble can be overcome by using an auxilliary or pre-combustion chamber, the hot gases from which are injected into the main chamber. This is run sufficiently long to initiate combustion in the main chamber and then switched off, thus conserving the plug for future use. A third method is to inject auxiliary fuels, hypergolic or self-igniting monergolic, into the chamber before the main fuels. It has been suggested that zinc diethyl, which is spontaneously inflammable in air could be sealed in the fuel line between bursting discs and thus be forced into the chamber before the main propellants.

Because of the high consumption of a rocket unit any rocket-propelled equipment needs to carry a very large amount of fuel. In order to keep the propellant volume, and thus structure weight and frontal area, to a minimum it is essential to use high density propellants. A convenient index which takes account of both specific impulse and density is the "density impulse." This is obtained by multiplying the specific impulse by the density of the propellant

mixture and is a measure of the impulse per unit volume. From the figures given for density impulse in Figure 2 it is seen that an otherwise promising propellant, liquid oxygen/liquid hydrogen, is one of the worst mixtures from this point of view, whereas the light metals show up even more favourably than when considered purely on the basis of specific impulse. Calculations have in fact shown that a small interceptor fighter would have a slightly shorter duration when using a liquid oxygen/liquid hydrogen motor than when using a liquid oxygen/methanol motor.

After examining the various propellant combinations in the light of the foregoing remarks a choice is finally made. It will be found that the final choice is lacking in many respects—often in all, and always from the point of view of ease of handling.

The Rocket Motor

The rocket motor can be divided into three separate parts as shown diagrammatically in Figure 1, namely the propellant injectors, combustion chamber and expansion nozzle. The functions of these three parts are closely interrelated but as far as possible I will consider the specific characteristics of the injector separately from the rest of the motor. The chief basic factors governing the motor design are good propellant mixing by the injectors; adequate combustion chamber volume to ensure completeness of reaction, combined with a minimum surface area to reduce heat transfer; adequate material strength at the working temperatures to withstand the pressure; good internal flow-pattern to prevent hot-spots (this is influenced to a great extent by the injectors); and lastly, light weight and ease of manufacture.

Reference to the expression given earlier shows that the specific impulse of a given propellant is dependent on the pressure ratio across the expansion nozzle. Figure 5 shows the variation of thermodynamic efficiency with pressure ratio and it can be seen that the increase becomes comparatively small with pressure ratios greater than about 40. It does not, therefore, normally pay to increase chamber pressures much above 40 atmospheres and, in point of fact, due to weight considerations the optimum value of chamber pressure at the present day is far below this; e.g. V.2. 15·5 atmos. and the Walter 109–509 21 atmos. As suitable lighter and stronger materials become available this optimum value will rise.

The combustion process itself can be broken down into several phases; the mechanical preparation and atomisation of the propellants by spraying; mixing of the substances; pre-heating up to the ignition temperature; and reaction. These processes overlap according to the construction employed. The mechanical atomisation can be accompanied by mixing and heating and even by partial reaction. From the point of view of heat transfer the chamber should be kept as small as possible so it is essential to investigate thoroughly the course of the reaction. Reactions proceed most rapidly when the components are in the gaseous state. It is thus essential to mix the components as intimately as possible and bring them to react in the gaseous state. Two methods are possible here, either pre-heat outside the combustion chamber to such a temperature that the components are gaseous and mix them before entry

to the chamber, or to inject and mix them in the liquid phase, the latter method being much the more common of the two.

The mechanism behind the spraying of liquids is that the pressure energy is set free on the flowing out of the fluid and gives a high velocity to the jet. By friction with the surrounding medium the jet breaks up into small particles. The shape of the outlet and density of the surrounding medium are important in deciding the degree of atomisation and depth of penetration. The higher the back pressure and the higher the pressure differential across the injector the smaller are the sprayed drops and the more quickly they are heated and evaporated. However, for quick heating it is not essential to have the droplets

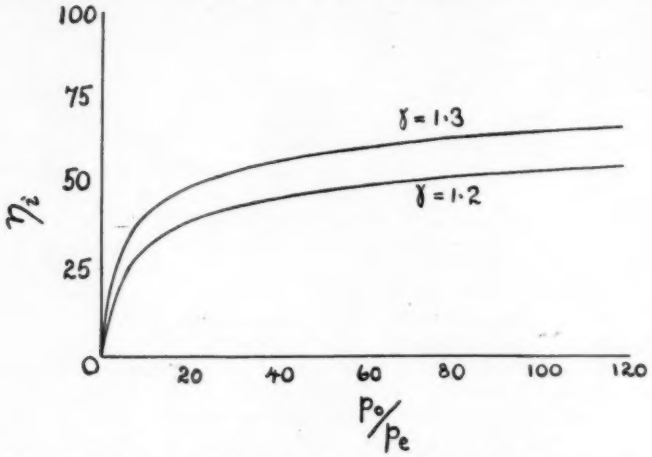

FIG. 5. Variation of ideal thermodynamic efficiency with pressure ratio.

uniformly small. The important part is that a certain number of very small droplets exist so that they evaporate rapidly and form reaction-centres within the mixture not yet completely heated. This penetration with heat centres can considerably increase the heating of the mixture. The heating of the unburnt mixture to the ignition temperature occurs before the flame-front, partly by radiation but chiefly by direct thermal conduction. The length before the flame-front is thus a function of the inflow velocity and the time required to heat the gases to ignition temperature. A temperature change occurs through the flame-front due to the reaction and in consequence of this an increase of volume occurs which causes a velocity change and hence a pressure difference; in the rest of the chamber the reaction is completed. Till now few determinations have been made of the combustion velocities or ignition velocities of rocket propellants at high pressures and hence determination of combustion chamber volume has to be done empirically.

For a given propellant mixture, combustion chamber pressure, injector type, chamber shape and throughput the time of stay in the chamber is given

by kV/A where V/A is the ratio of combustion chamber volume to throat area and k is a constant. V/A is known as the characteristic length L* but again has not been investigated for many propellants. The usual method is to base the value of L* on that of an existing successful chamber using the same propellants under the same conditions.

To return to injectors, most work has been empirical and consequently many forms have been devised, of which the swirl-nozzle is probably the best known. Here the liquid flows tangentially into a nozzle chamber and on ejection forms a very thin rotating cone. The larger the amount of energy of the rotating ring in the nozzle chamber the thinner the cone and the smaller the drops.

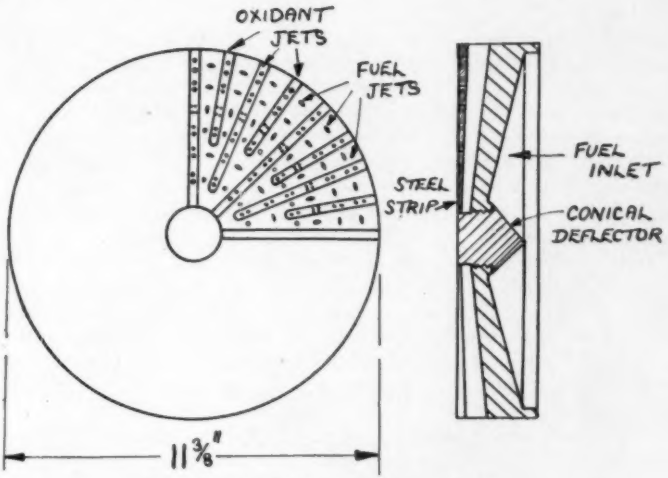

FIG. 6. Wasserfall burner plate—jet type.

In order to get as near as possible homogeneous mixing many injectors have multiple injection points for the propellants, as in the V.2. type. Here liquid oxygen is injected from a rose from a multitude of plain holes and alcohol is injected toward these streams from a large number of swirl type holes; eighteen of these units are used in V.2. although due to the complexity of the system a single ring-type injector was being developed to handle the whole flow. This consists of a cast plate with a series of ring-slits in it, behind which is fixed another plate containing annular spaces corresponding to the front slits. The fuel and oxidant are sprayed through alternate slits. Another example was designed for use on the flak rocket "Wasserfall." An earlier type of plate injector for the same unit is shown in Figure 6. In this type the propellants are injected through series of radial holes inclined at a small angle to one another. In all of the injectors so far described, with the exception of the V.2. type, one of the main drawbacks has been the comparatively small angle

at which the fluids meet. When starting the unit this means that, with the lack of back pressure and the small force with which the streams meet the mixing is very poor or in extreme cases non-existent. This means, particularly with non-hypergols, that sufficient of the propellants may collect before ignition to cause an explosion. There are two ways of overcoming this poor initial mixing either by having the jets inclined at an obtuse angle to each other or by using a target plate. In the latter method advantage is taken of the fact that a reaction always commences most readily at a surface, and consequently a target plate is used at which the propellants are aimed, the mixing and ignition taking place on the surface. This latter method is of chief interest for "one-shot" rockets as the target plate burns away during running, or in rockets using low specific impulse propellants as in the case of the Walter 109–500 "cold" assisted take-off unit.

And now for the question of cooled or uncooled motors. A rocket motor operates under more severe conditions of high temperature and continuous heat release than any other heat engine. The heat liberated per unit volume per unit time is about four times as great in a rocket with a specific impulse of 200 secs. as in a modern high temperature furnace, that is about 250,000,000 C.H.U./ft.^{3}/hr. The chamber temperatures obtained with present day fuels vary between 1,500°C. and 2,500°C. for most practical units. Motors may be broadly classified under two headings depending on whether the heat transferred from the gases is absorbed by the chamber material ("uncooled" type) or by all or part of the propellant ("cooled" type). The cooled type can be further sub-divided into the type in which the coolant is circulated around the chamber and nozzle through a duct before injection, known as "regenerative" cooling, and the type in which part of the coolant is injected directly into the chamber, usually through rows of small orifices drilled through the inner wall, in such a way as to provide a coolant film on the inner wall surface ("film-cooled" type).

The uncooled motor is usually made of metal with sufficient mass to absorb the transmitted heat for the firing period, normally less than 60 secs. With special low temperature propellants, for example, hydrogen peroxide as used in the 109–500 assisted take-off unit which produces a temperature of 500°C., quite a light metal construction suffices and the unit can be run for an unlimited period. With higher performance propellants it is usually necessary to use a metal with a high heat capacity and high conductivity in order successfully to absorb the heat, or, since these materials mostly have low melting points (e.g. copper, aluminium), an alternative is to use a material such as a ceramic or carbon with a high melting point, although possessing a much lower heat capacity and conductivity. The chief drawbacks here are that ceramics crack very badly due to thermal stresses and that carbon erodes very badly due to the scrubbing action and oxidation at the throat. The necessity for providing sufficient heat capacity renders high performance uncooled motors so heavy as to limit their usefulness to applications of the shortest duration, thus interest centres chiefly on cooled motors.

It is not immediately obvious whether in a particular case the propellant flow can absorb the heat transmitted to the walls and remain in the liquid

phase. It is very often possible to do this but a marked scale effect exists for cooling. The rate of coolant flow increases linearly with motor thrust, whereas the area to be cooled increases less rapidly. This means that for certain critical cases a given propellant can be used to cool a large motor but not a small one. For the cooling of the walls it is first necessary to know what heat quantities will enter the wall. Within the combustion chamber, where the gases are almost stationary an appreciable part of the heat will be transferred to the wall by radiation. However, towards the nozzle the gases get into motion and then cool adiabatically; at the throat they attain a stream velocity equal to the

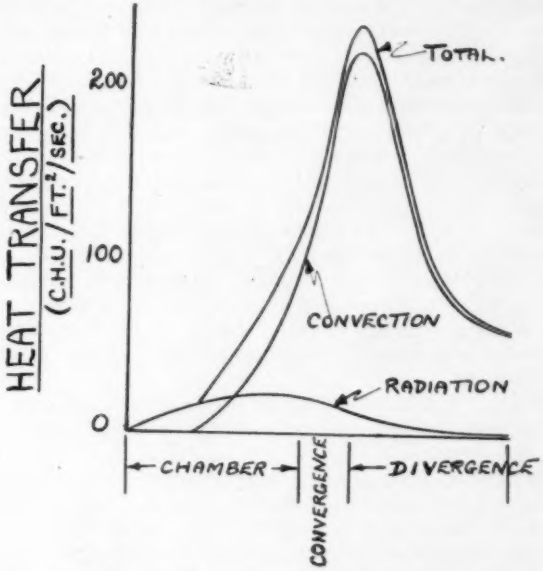

FIG. 7. Heat transfer in typical rocket motor

speed of sound in the gas, and in the divergent part of the nozzle they finally flow at very high velocities (about 7,000 ft./sec.) and are considerably cooler than in the chamber. Figure 7 gives an indication of the relative amounts of heat transfer due to radiation and convection throughout the motor. It can be seen that radiation is important only in the chamber and is negligible in the nozzle, and the convection increases very rapidly in the nozzle due to the rapid increase in stream velocity. As propellants with higher performances are used the cooling problem will become more acute and various steps will have to be taken to solve it. The development and use of materials with high melting points and low thermal conductivity will be one major contribution. Reduction of the area in contact with the gases has already been mentioned in connection with the combustion chamber and this can also be done, to a certain

extent, in the nozzle. The most critical portion of the nozzle is the throat and the divergent part near the throat. The included angle of the divergent part in modern motors is 30° or less, and a further method of reducing heat transfer is by increasing this angle, when the nozzle surface will be reduced. This increase in angle will mean a decrease of nozzle efficiency due to the increased radial momentum component of the main gas stream but this will probably be more than offset by the higher allowable chamber temperatures. The total amount of heat that can be safely transmitted to the coolant is fixed for a given coolant flow as is the maximum allowable coolant film temperature.

A promising means for controlling the heat flow to the wall of a motor is by the method of film cooling mentioned earlier. This method consists of introducing small quantities of coolant at many points distributed uniformly over the interior surface. The coolant so introduced is spread over the wall in a thin film and eventually evaporated, the essential advantage being that the vaporisation of the screening film increases its heat absorbing capacity many times over that of a system in which the coolant remains in the liquid phase. It has the further advantage that the heat does not have to be conducted through the walls. A successful example of a rocket using film-cooling is V.2. In this about 3 per cent. of the total mass-flow is diverted to the cooling film and about half of this actually takes part in the combustion and hence is not wasted. Four rings of holes are used for injecting the coolant and these were determined purely by experiment. A further possibility in the film-cooling line is the use of sintered metal nozzles. These can be made porous so that coolant can be forced through and so give a uniform internal cooling film.

In rockets for continuous propulsion of aircraft, particularly fighter and interceptor aircraft, it is desirable to be able to cruise at a reduced thrust in order to conserve fuel. The obvious answer is to cut down the propellant supply to the motor, but this has serious drawbacks. In the first place the pressure ratio is reduced and hence the specific impulse reduced, thus offsetting to some extent the decrease in consumption due to decreased thrust. Secondly, combustion becomes worse as the chamber pressure is decreased and combustion may be less complete, again decreasing the specific impulse. Also the roughness of combustion sets a lower limit on the minimum allowable thrust. One method of overcoming this difficulty is by using several chambers, as in the American 1500 N4C, liquid oxygen/alcohol, 6,000 lb. thrust, 4–chamber motor, and running as many as may be necessary to give the required thrust or as in the Walter 109–509c, to use a special small chamber for cruising purposes. Another method is to vary the throat area with the propellant flow so as to maintain a constant pressure ratio. This could probably be achieved by means of a movable bullet in the throat similar to the bullets used in the German turbojet units. The chief difficulties introduced by this scheme are the extra surface introduced into the gas stream and requiring cooling, and the problem of sealing the moving part at the front of the chamber.

The application for which a motor is designed also has a profound effect on its design. The major variables are, magnitude and duration of thrust; fixed

or variable thrust; whether for repeated use or a "one-shot," and in the former case its total operating life. The lower limit of thrust for which motor manufacture is simple is of the order of 100 lb., below this figure the injection orifices become so small as to be subject to clogging. There also seems to be a slightly higher limit below which maximum combustion efficiency is not attained. This is because, besides a certain minimum value of L* being required for complete combustion, a certain minimum chamber length is also required. At the other end of the scale, however, there seems to be no fundamental upper limit to the size of a rocket motor; the largest so far built is the German V.2. delivering about 55,000 lb. thrust, whilst Sänger in Germany was developing a 225,000 lb. liquid oxygen/oil rocket, and the A.10. which was to have been used as a booster unit in conjunction with a modified winged V.2. was to have delivered 440,000 lb. Dr. Malina of the California Institute of Technology considers that a motor with a chamber diameter of 12 ft. and a throat diameter of 5 ft. capable of delivering a thrust of 1,000,000 lb. is a present-day engineering possibility.

After a motor has once safely reached a state of thermal equilibrium the duration of continuous operation is limited only by nozzle erosion or amount of propellant available, usually the latter in a cooled motor. For assisted take-off and aircraft propulsion then, the duration is limited only by the amount of fuel which can be carried whereas in a wingless projectile the duration is obviously limited by the fact that the weight of propellant must be less than the magnitude of the thrust. We can, therefore, draw up a list of approximate durations of motor operation typical of specific applications as follows:—

Aircraft assisted take-off	10– 60 secs.
Aircraft superperformance	30–300 secs.
Aircraft propulsion	5– 60 mins.
Projectile propulsion	10–100 secs.

The total operating lifetime of a motor is usually limited by nozzle erosion. A liquid cooled nozzle can with present fuels last for several hours, 109–509A.2. chambers have been run for 1½ hours without failing and even a V.2. motor designed to run for a maximum of 60 secs. has been run for over an hour in bursts of 50 secs. at a time, with no appreciable damage to the throat. In expendable service, however, it is often possible to tolerate a certain amount of erosion and build a much lighter motor than would be allowable for repeated operation, and if a human operator is not required, safety factors can be reduced, thus even further reducing the total weight.

Fuel Supply Systems

In most other heat engines the fuel flow is so low that the fuel feed systems are very small and light compared with the rest of the engine. In rockets, however, the propellant flow rate is so high that the feed system occupies a high percentage of the total weight of the unit, and this being the case great efforts are being made to keep the feed system as light as possible. The main systems in use are the mechanically driven pump and gas pressure. This latter method is further divided into systems carrying their own gas and those using a gas generator. The simplest feed system is that using a compressed gas and this

method has so far been the most widely used, being most suitable for units having durations of less than approximately 60 secs. The gas generator system has not yet got past the experimental stage, but does seem to be very promising for future applications due to the low weight installations possible. Again there are several sub-divisions of this method, the chief ones being a slow-burning cartridge of solid fuel—usually cordite, a generator using the main propellants or a generator using subsidiary fuels. Any of these methods can be used in conjunction with stored gas in order to reduce temperatures.

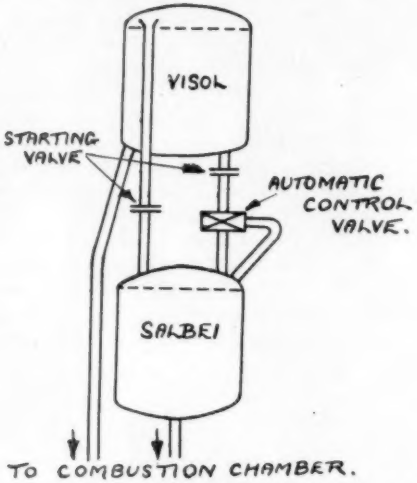

FIG. 8. Feed system using main propellants.

For mechanically driven pumps the only practical methods of operation are to drive them by turbines or, in the case of superperformance and assisted take off units, from the main aircraft engines. For the turbine driven types so far evolved a separate gas generator has been used, as for example in the 109–509. Here hydrogen peroxide is tapped off from the outlet of the pump and fed through a catalyst chamber producing a mixture of oxygen and steam at about 520°C. which is then passed into a single-stage turbine. This system requires starting, either by an electric motor or by an auxiliary supply of hydrogen peroxide. Another method not so far developed is to tap off a small amount of combustion gas from the chamber and to use it in a gas turbine driving the pumps.

In the gas generator method many schemes have been devised but I intend to deal only with two basic types, the slow-burning cartridge and the use of one or more of the propellants. An interesting example is the system under development for the Wasserfall flak rocket. The propellants used are nitric acid and visol (a mixture of vinyl compounds). The system is shown in Figure 8. The two propellant tanks are connected by a pipe fitted with a

starting valve and an automatic control valve. Another pipe connects the ullage spaces, i.e. the empty spaces at the tops of the tanks. To operate the system both starting valves are opened, fuel flows into the acid tank and the gas so produced passes to the ullage spaces thus further increasing flow. The pressure rises to a preset valve when the automatic control valve limits the flow of fuel and maintains a steady pressure in the tanks. A slow burning cartridge device was also under development for Wasserfall and is shown in Figure 9. The device consists of a steel cylinder, containing the charge, connected to a steel metering choke and insulated at the end by a ceramic

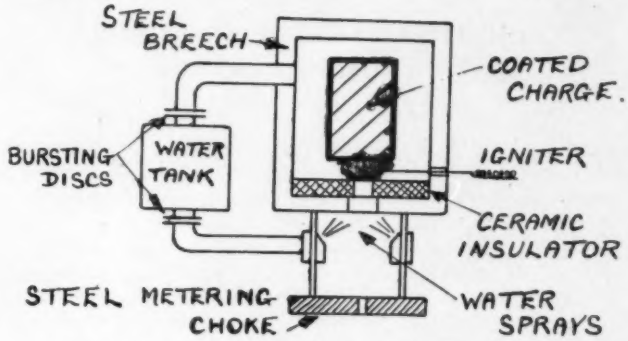

FIG. 9. Feed system using slow-burning cartridge.

plate. A small water tank is also connected to the breech, discharging water through sprays into the gas stream, thereby cooling the gases to the desired temperature. Compared with the original nitrogen fuel feed system weighing 704 lb. this slow burning cartridge unit weighs only 176 lb.

Conclusions

First, in the line of increase of motor performance, there appears to be little hope of any startling increase by any increase in the efficiency of nozzles, chambers or injectors, as the efficiencies of these are already very high. We are left as the only alternative the use of better fuels. All fuels containing H, C, O and N produce combustion products which dissociate so rapidly above 2,500°C. that the temperature at which equilibrium is reached is definitely limited to values of less than 3,500°C. Since this limit is imposed by the products of combustion rather than by the latent energy of the propellant in its liquid state it seems unlikely that specific impulses of much more than about 300 secs. will be obtained from this class of propellant. The possibility of higher specific impulses seems to lie in the propellants mentioned in Figure 2 and investigation of these would be of interest.

If propellants with higher specific impulses are introduced the cooling problems will become more acute and development of materials with greater strength at high temperatures and with low heat conductivity will be required. Ceramics seem to be the most likely materials in this field. Further investigation

is also required into film cooling, particularly with regard to porous metals. Reduction of weight, of chief interest in projectiles, will be required in all parts of the rocket unit and one very promising means of doing this, at any rate within a certain impulse range, seems to be the development of gas generators for fuel feed.

In conclusion I would like to express my thanks to the Ministry of Supply for allowing me to use much of the material contained in this paper and to state that the views expressed are my own and do not necessarily reflect those of the Ministry or any other Government department.

ZUR THEORIE DER RAKETEN*

(ON THE THEORY OF ROCKETS)

By Professor J. Ackeret†

Now that rockets of some size have at last successfully travelled to a great height, the interest in this principle of propulsion—quite apart from the military point of view—has increased considerably. Further development has shown satisfactorily that this principle affords the only possibility of space travel beyond the atmosphere. From the earlier investigations, especially those of Oberth,[1] it is clearly obvious that the energy required lies at the extreme limit of that which is attainable with the aid of our most powerful fuels. It is understandable, therefore, that the opening up of new sources of energy, such as nuclear energy on an industrial scale, has revived new hope for space travel. A discussion of rocket theory from the point of view that energy is available of a different order of magnitude than of heretofore may therefore not be out of place.

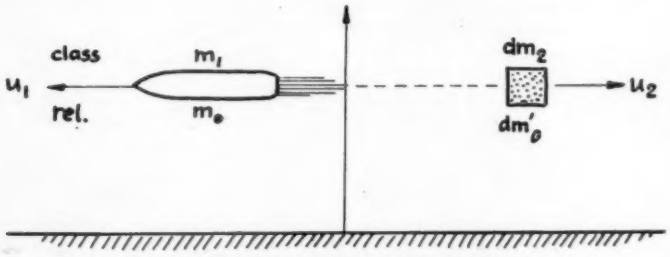

Fig. 1. Masses and velocities of rocket and jet.

I. Elementary Rocket Theory

Consider a rocket in free space with no external force such as gravity, etc., acting upon it. The behaviour of the system is observed from the centre of gravity of the system as origin. A mass is ejected from the rocket with a relative velocity "v" directed towards the right.

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† Translated by L. Gilbert, B.Sc., A.R.C.S.

After a time "t" let the mass of the rocket be "m_1" and the velocity "u_1" directed towards the left; let the ejected mass "dm_2" have a velocity "u_2" directed towards the right.

Now,

(a) from conservation of mass,

$$dm_2 = - dm_1 \qquad \ldots\ldots\ldots (1)$$

(b) from conservation of momentum,

$$d(m_1.u_1) = dm_2.u_2 \qquad \ldots\ldots\ldots (2)$$

(c) from the theorem of compounding of velocities,

$$u_2 = v - u_1 \qquad \ldots\ldots\ldots (3)$$

Eliminating dm_2 and u_2,

$$m_1.du_1 = - dm_1.v \qquad \ldots\ldots\ldots (4)$$

Integrating,

$$\log m_1 = - u_1/v + \text{constant}.$$

Let the mass (m_1) at the start ($u_1 = 0$) be M_o; then the well-known equation of rocketry is obtained,

$$m_1/M_o = \exp.\ (- u_1/v).$$

At the all-burnt stage let the final mass of the rocket be "M_t" and its velocity "U." Let the mass-ratio M_o/M_t be "W." Then,

$$W = M_o/M_t = \exp.U/v \text{ or } U/v = \log W \qquad \ldots\ldots (5)$$

It is seen from this that a mass ratio of $W = e = 2{\cdot}718$ is necessary for the final velocity (U) of the rocket to be equal to the jet velocity (v).

Theoretically, a jet velocity of 4450 m/sec. is attainable with petrol (C_8H_{18}) and oxygen, and of 5670 m/sec. with hydrogen and ozone (2). In actual practice, a jet velocity of 4000 m/sec. can be counted on at the best. Therefore in order to reach the velocities of 20,000–25,000 m/sec. which are necessary for a lunar flight including landing, a very high mass ratio must be aimed at. Such end velocities would only be possible with so enormous a rocket as to lie outside the realms of practical construction.

II. Relativistic Rocket Theory

In order to calculate precise formulae from extremely high jet velocities, it is necessary to employ relativistic equations. In this case, the principle of equivalence of mass and energy is used in place of conservation of mass and the rest masses 'm_o" and "dm'_o" are introduced.

(a) from equivalence of energy,

$$d\left\{\frac{m_o c^2}{\sqrt{1 - u_1^2/c^2}}\right\} = - \frac{dm_o.c^2}{\sqrt{1 - u_2^2/c^2}} \qquad \ldots\ldots\ldots (6)$$

(b) from conservation of momentum,

$$d\left\{\frac{m_o.u_1}{\sqrt{1 - u_1^2/c^2}}\right\} = \frac{dm_o.u_2}{\sqrt{1 - u_2^2/c^2}} \qquad \ldots\ldots\ldots (7)$$

(*c*) velocity transformation,

$$u_2 = \frac{v - u_1}{1 - \frac{u_1 v}{c^2}} \qquad \ldots\ldots\ldots (8)$$

where "v" is the jet velocity relative to the rocket.

Eliminating dm_o' and u_2 as before, (6) and (7) give,

$$dm_o \cdot \frac{c^2}{\sqrt{1 - u_1^2/c^2}} + m_o c^2 . d\left\{\frac{1}{\sqrt{1 - u_1^2/c^2}}\right\} = - dm_o \cdot \frac{c^2}{\sqrt{1 - u_2^2/c^2}} \qquad \ldots\ldots (9)$$

$$dm_o \cdot \frac{u_1}{\sqrt{1 - u_1^2/c^2}} + m_o \cdot \frac{du_1}{\sqrt{1 - u_1^2/c^2}} + m_o u_1 . d\left\{\frac{1}{\sqrt{1 - u_1^2/c^2}}\right\} = dm_o \cdot \frac{u_2}{\sqrt{1 - u_2^2/c^2}} \qquad \ldots\ldots (10)$$

The application of (8) gives,

$$\frac{c^2}{\sqrt{1 - u_2^2/c^2}} = \frac{c^2 - u_1 . v}{\sqrt{1 - v^2/c^2}\sqrt{1 - u_1^2/c^2}} \qquad \ldots\ldots (11)$$

$$\frac{u_2}{\sqrt{1 - u_2^2/c^2}} = \frac{v - u_1}{\sqrt{1 - v^2/c^2}\sqrt{1 - u_1^2/c^2}} \qquad \ldots\ldots (12)$$

dm_o' is eliminated from (9) and (10) by using (11) and (12),

$$m_o \left\{\frac{c^2 - u_1 . v}{\sqrt{1 - u_1^2/c^2}}\right\} du_1 + m_o v(c^2 - u_1^2) . d\left\{\frac{1}{\sqrt{1 - u_1^2/c^2}}\right\} + dm_o \cdot \frac{v(c^2 - u_1)}{\sqrt{1 - u_1^2/c^2}} = 0 \qquad \ldots\ldots (13)$$

Rearranging, etc.,

$$\frac{dm_o}{m_o} = - \frac{du_1}{v(1 - u_1^2/c^2)} \qquad \ldots\ldots (14)$$

Integrating,

$$\log m_o = - \frac{c^2}{v} \int \frac{du_1}{c^2 - u_1^2} \text{ constant} \qquad \ldots\ldots (17)$$

$$= \frac{c^2}{v} \cdot \frac{1}{2c} . \log \frac{c + u_1}{c - u_1} \text{ constant} \qquad \ldots\ldots (15)$$

With an initial mass $m_o = \mathrm{M}_o$ and a final mass $m_o = \mathrm{M}_t$ and $u_1 = \mathrm{U}$, then the relativistic equation for rockets is given by,

$$\mathrm{W} = \frac{\mathrm{M}_o}{\mathrm{M}_t} = \left\{\frac{1 - U/c}{1 + U/c}\right\}^{-c/2v} \qquad \ldots\ldots\ldots (16)$$

$$\text{or} \qquad \frac{\mathrm{U}}{c} = \frac{1 - \mathrm{W}^{-2v/c}}{1 + \mathrm{W}^{-2v/c}}$$

It can be quite simply shown that in the limiting case of $c \to \infty$, (17) transforms to (5).

In Figure 2 values of U/c against $^1/_W$ are plotted for $v/c = 1, \frac{1}{2}, 1/10$, as well as for the classical case.

From Figure 2 it is seen that,

(*a*) for $v = c$ (photons) departure from the classical theory appears for relatively low mass ratio, while for $v = c/10$ classical theory is quite sufficient even for a very high mass ratio.

(*b*) from relativity theory, the final velocity U can obviously never exceed the velocity of light c.

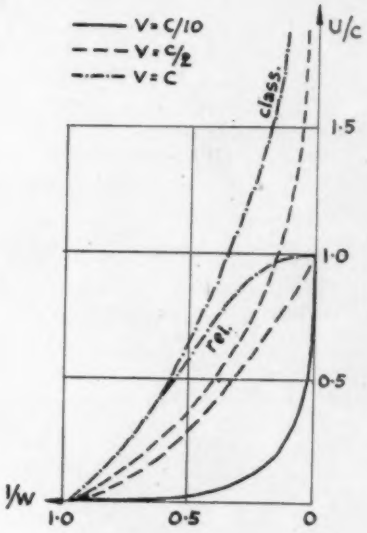

FIG. 2. Final velocities of the rocket as a function of the mass ratio W for different jet velocities. For $v = c/10$ there is no practical difference between the classical and relativistic approach.

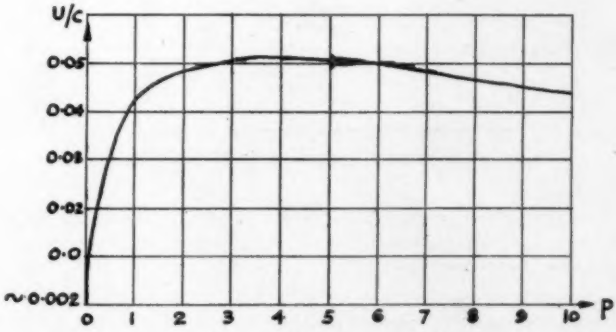

FIG. 3. Final velocity as a function of the factor "P."

III. The Application of Nuclear Energy to Rocket Propulsion

The jet velocity "v" may be calculated in the following manner.

The final mass M_t is fixed at a predetermined value. Let αM_t be the mass of the active substance which evolves the energy and is not ejected in the exhaust Let the free energy transformed and emitted be the fraction $\delta\alpha M_t$ corresponding to the mass defect between the initial and final states. The energy evolved is therefore given by

$$\delta\alpha M_t c^2$$

δ is less than 1 per cent. even in the most powerful nuclear reactions known.

For uranium fission	$\delta = 0{\cdot}0731$ per cent.
For $Li^7 + H^1 \rightarrow 2\,He^4$	$\delta = 0{\cdot}232$ per cent.
For $4\,H^1 \rightarrow He^4$	$\delta = 0{\cdot}715$ per cent.

Associated with this active, energy-producing mass is a "passive" mass "$p.M_t$" which is ejected.

The total initial mass is therefore,

$$M_t + \delta\alpha M_t + p.M_t = M_t(1 + \delta\alpha + p) = M_o$$

Whence,

$$W = \frac{M_o}{M_t} = (1 + \delta\alpha + p) \qquad \ldots\ldots (18)$$

The principle of equivalence of mass and energy applied to the system of co-ordinates of the rocket gives,

$$\text{kinetic energy} = \frac{M_t c^2 . p}{\sqrt{1 - v^2/c^2}} - M_t c^2 . p$$

which is developed by the released energy $\delta\alpha M_t c^2$.

Thus,

$$p\left\{\frac{1}{1 - v^2/c^2} - 1\right\} = \delta\alpha$$

$$\sqrt{1 - v^2/c^2} = \frac{1}{1 + \delta\alpha/p}.$$

$$v/c = \sqrt{1 - p^2/(p + \delta\alpha)^2} \qquad \ldots\ldots (19)$$

The rocket equation therefore reads,

$$U/c = \frac{1 - \left\{\frac{1}{1 + \delta\alpha + p}\right\} 2\sqrt{1 - p^2/(p + \delta\alpha)^2}}{1 + \left\{\frac{1}{1 + \delta\alpha + p}\right\} 2\sqrt{1 - p^2/(p + \delta\alpha)^2}} \qquad \ldots\ldots (20)$$

$\delta\alpha$ must now be evaluated. The question arises as to what is the optimum value of "p." It is assumed that it is possible, in general, to impart the energy $\delta\alpha M_t c^2$ to a small arbitrary mass pM_t.

For the factor $\delta\alpha = 0{\cdot}2$ per cent. (quite a high value), a plot of "p" against "U/c" is given in Figure 3, from which it can be seen that $p \sim 4$ would give the greatest velocity. For $p = 0$, the final velocity of $U = 0{\cdot}002c$; furthermore

the rest mass $\delta\alpha M_t$ gives a reaction $U/c \sim \delta\alpha$ in the form of radiated photons ($v = c$).

In Figure 4 v/c is plotted against U/c for a mass ratio of 5, which is quite a reasonable practical value. Note that classical theory gives a straight line.

Note that in Figure 4 in the neighbourhood of the origin the relativistically calculated curve cannot be distinguished from the classical case.

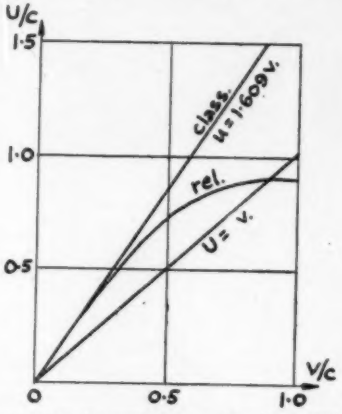

FIG. 4. Final velocity of the rocket as a function of the jet velocity for a mass ratio of 5.

Classically, U/c may be calculated very simply as follows.

$$\delta\alpha M_t c^2 = \tfrac{1}{2}\, p.M_t v^2$$

$$v/c = \sqrt{2\delta\alpha/p} \qquad \ldots\ldots (21)$$

and from (5)

$$W = p + 1$$

$$U = v.\log\,(p + 1) \qquad \ldots\ldots (22a)$$

$$U/c = \sqrt{2\delta\alpha/p.\log\,(p + 1)} \qquad \ldots\ldots (22b)$$

This method gives almost complete agreement with the curve in Figure 4 with the exception of "p" = 0, when the limiting value of 0 from (22b) is obtained.

For larger values of "p," (22b) may be used directly.

The actual composition of the ejected matter is of little importance for the moment.

With $\delta\alpha = 0{\cdot}2$ per cent. and "p" = 4 a velocity of 5 per cent. of that of the velocity of light would be possible, which would certainly be sufficient for all known planetary journeys. Even this value of $U/c = 0{\cdot}2$ per cent. developed with photons alone ($U = 600$ km/s.) would be very exceptional as may be seen from the fact that the escape velocity in the neighbourhood of the sun has a value of 617 km/sec. However, the intrinsic difficulty now becomes apparent insofar as with the materials now available, it scarcely appears possible to produce a velocity in accordance with (19) or (21) with a small value of "p."

Let us now consider a few of the possibilities.

(*a*) PHOTONS. (Radiation Pressure).

It should not be forgotten that the radiation pressure developed by a beam of radiation is quite insignificant in amount and this is very much higher than in any lamps, searchlights, X-ray tubes, etc. If, for example, the reaction is calculated which is produced from a parabolic mirror at whose focus is a spherical black body of 10 cm. radius at 4,000°K., then a quite insufficient thrust is found, i.e. the calculated final velocity would be obtained ultimately, but only after a very long time indeed. So long as the space-ship is in the Earth's gravitational field, then the forces employed must be at least of the same order of magnitude as the mass involved. Radiation pressure therefore appears to be impracticable.

(*b*) CHARGED PARTICLES.

Charged particles of either sign, e.g. electrons, cannot be considered owing to the rapid charging up of the rocket. Even the smallest quantity ejected would lead to the cessation of emission almost immediately.

(*c*) UNCHARGED PARTICLES.

Uncharged particles of high velocity cannot really be controlled without the surrounding walls attaining the temperature equivalent to that velocity; furthermore, very high intensities must be employed. It is assumed also, that only by means of the production of heat andthe ejection of the hot "inert" exhaust, can a sufficiently powerful jet of matter be generated. Then, however, the well-known laws applying to the ejection of gases into a vacuum apply. For a gas at a temperature "T" in the rocket combustion chamber, with specific heats Cp and Cv per gram (assumed constant) and of molecular weight "m,"

$$v = \sqrt{2\,\mathrm{J.C}_p.\mathrm{T}}$$

$$\text{and } \mathrm{C}_p = \mathrm{C}./m \; ; \quad v = \mathrm{K}\sqrt{\mathrm{T}/m}$$

J = mechanical equivalent of heat; $\mathrm{K} = \sqrt{2\,\mathrm{J.C}}$. is constant for gases of equal atomicity.

$$\mathrm{C} \begin{cases} \text{for monatomic gases} = 5\mathrm{R}/2 = 5 \text{ cal./g.mol.} \\ \text{for diatomic gases} = 7\mathrm{R}/2 = 7 \text{ cal./g.mol.} \end{cases}$$

If the undoubtedly high value of 4,000°K. is assumed for "T," then it follows that for,

hydrogen (monatomic	$v =$	12,900 m/sec.
(diatomic)	$v =$	10,800 ,,
helium (monatomic)	$v =$	6,450 ,,
nitrogen (diatomic)	$v =$	2,900 ,,

"p" may also be evaluated. The energy $\delta\alpha \mathrm{M}_t c^2$ depends on the temperature of the mass $p\mathrm{M}_t$ from almost zero to T°K.

$$p = \frac{\delta\alpha c^2}{\mathrm{J.C}_p.\mathrm{T}} \qquad \ldots\ldots\ldots (24)$$

thus, for diatomic hydrogen for example, with $\delta\alpha = 0{\cdot}2$ per cent.

$$p = 3{\cdot}10^6$$

which is an impossibly high value.

It is seen at once that it is not possible to utilise the available energy corresponding to a $\delta\alpha = 0{\cdot}2$ per cent. Thus the high velocities (15,000 km/sec.) are apparently unattainable because we cannot handle the great energy involved, at least not if it involves the production of heat.

If $p = 100$ is used as the basis of calculation (this being perhaps the maximum conceivable practical value), then $\delta\alpha = 6{\cdot}5.\ 10^{-8}$ is sufficient, i.e. $\alpha = 10^{-4}$ and $\delta = 0{\cdot}07$ per cent. (U235). Then from (22a) the velocity is,

$$U = 10{,}800 \log 101 = 50{,}000 \text{ m/sec.}$$

This would suffice directly for a lunar journey and even for a journey to Venus or Mars.

A mass ratio of $p = 10$ gives,

$$U = 10{,}800 \log 11 = 26{,}000 \text{ m/sec.}$$

and would be sufficient for a lunar journey.

The results of the discussion can be summarised as follows,

1. The energy resulting from the nuclear fission of U235 would be sufficient for space travel in the solar system.
2. The difficulties are,
   (*a*) in the production of the high jet velocities by thermal transfer of energy.
   (*b*) in the technical realisation of a sufficiently high mass ratio.

If the forgoing considerations on space-travel are compared with chemical fuel considerations, then the only possible advancement lies essentially in the attainment of a jet velocity 2 or 3 times higher by using hydrogen as an inert propellant. For the rest, the technical problems are very much the same.

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(1) H. Oberth. "Die Rakete zu den Planetanraum." Munchen, 1923. E. Sanger. "Raketenflugtechnik." Munchen, 1933.
(2) Sanger, loc. cit., p. 66.

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ROCKET FORMULAE

By H. E. Ross and R. A. Smith

The following formulae and transpositions will be found useful in extracting a basis of comparison from the various forms of data of rocket performance in common use.

DEFINITIONS.

Thrust. The total recoil force produced by the rocket.

Total Impulse. The thrust multiplied by the duration of thrust.

Specific Impulse (or *Specific Thrust*). The thrust produced for every unit ($m \times$ g.) of fuel discharged in unit time. (Note: The S.I. is sometimes abbreviated and given simply as so many seconds.

Specific Fuel Consumption. The quantity of fuel ($m \times$ g.) consumed in unit time for every unit of thrust produced. (Note: The Specific Fuel consumption is sometimes given in terms of grams per kilogram per second and sometimes as lbs. per pound per thrust hour.)

Rate of Fuel Discharge. The quantity of fuel ($m \times$ g.) discharged in unit time.

TOTAL WEIGHT OF FUEL $M_o - M_t$

Symbols:

Total Impulse $= I_t$
Thrust $= T$
Specific Impulse $= I_s$
Specific Fuel Consumption $= C_s$
Rate of Fuel Discharge $= D_r$
Exhaust Velocity $= v_e$
Acceleration due to gravity $= g$ (981 cm.sec.^{2} or 32·2 ft./sec.^{2})
Duration of discharge $= t$
Total weight ($m \times$ g.) of fuel $= Q$

TOTAL IMPULSE (I_t) =
$$tD_rI_s = tD_rv_e/g = tD_r/C_s = tT = v_eQ/g$$

THRUST (T) =
$$D_rI_s = D_rv_e/g = D_r/C_s = I_t/t = I_sQ/t$$

SPECIFIC IMPULSE (Specific Thrust) (I_s) =
$$T/D_r = v_e/g = 1/C_s = I_t/Q = tT/Q$$

SPECIFIC FUEL CONSUMPTION (C_s) =
$$D_r/T = 1/I_s = g/v_e = Q/I_t = Q/tT$$

RATE OF FUEL DISCHARGE (D_r) =
$$T/I_s = TC_s = Q/t = QT/I_t = gT/v_e$$

EXHAUST VELOCITY (v_e) =
$$gT/D_r = gI_s = g/C_s = gI_t/Q = gtT/Q$$

TOTAL WEIGHT OF FUEL (Q) =
$$tT/I_s = C_sI_t = tD_r = D_rI_t/T = gI_t/v_e$$

REVIEW

Ballistics of the Future

(With Special Reference to the Dynamical and Physical Theory of Rocket Weapons)

(Techn. Publ. Co., H. Stam, Haarlem, Holland, 1946, 472 pp., ill., index, 30 guilders = ca. £3. Written in English.)

This book was written during and after the war by two eminent Dutch scientists, Dr. Tr. J. H. J. Kooy, of the Aeronautical School at the Hague, and Prof. Dr. Tr. J. W. H. Mytenboogaart, Professor of the Technical University of Delft. The book is a technical work which covers practically the whole field of rocket ballistics.

It starts (Chapter I) with vector calculus and follows in Chapter II with vector calculus applied to general dynamics. Then follows two useful Chapters on the dynamics of the solid and on the equations of a connected system, giving among other things the velocity and kinetic energy of a system, as a function of general coördinates. Chapter V deals with the numerical integration of several kinds of differential equations of first and higher orders.

Chapter VI, on the projectile as a material point, provides the ballistic equation of a projectile moving in a vacuum and a gravitational field, both with and without air resistance, influence of wind, etc.

The most important part of the book is grouped under the headings:

I. Terrestrial dynamics of the rocket.
II. The German Robot-plane V1.
III. The German Giant Rocket V2.
IV. Extra-terrestrial dynamics of rockets.

Here are detailed considerations of rocket-performance, equilibrium, control and motion, and also reference to a rocket controlled by a gyro pilot. Then follows reference to the V1, with several technical drawings of the electric control system and auto pilot compass. After a consideration of the thermodynamics of the rocket-motor and its propellants, there follows a detailed technical description of the V2 and launching sites, illustrated with many previously unpublished drawings and photos.

The book is concluded with a treatment of a rocket flight around the moon and its method of calculation. Several folding maps are included.

No computation of the step-rocket is included and there is no mention of the technical applications of the rocket, such as assisted take-off units, bazookas, V2 developments, etc.

The standard of the entire work is high, rather more theoretical than technical, and is meant only for the reader who is well versed in the higher branches of mathematics. It gives a deep impression of the future possibilities of the rocket and can be strongly recommended to everyone interested in this field of science.

Ir. G. de Koningh.

CORRESPONDENCE

CHALLENGE OF THE SPACESHIP

SIR,

I have read with much interest the report of A. C. Clarke's paper and the subsequent discussion appearing in your Journal. May I venture to comment thereon.

Being fully conscious of the interest, the attraction and the value to human knowledge of Space Travel, I would say nothing to discourage it. I do, however, forsee dangers which are I fear insufficiently appreciated, and are moral rather than physical.

The paper contained a warning against the dangers of Imperialism which, I am glad to see, was generally approved. But the danger forseen was, it would appear, of national Imperialism. Surely World Imperialism might well be equally or rather more dangerous. Both the lecturer and those joining in the discussion envisaged men settling down in other worlds, at first in protected cities, later through adaptation throughout these worlds. Is not this just the old story of the colonisation of America, Africa and Australasia? First, the explorer, then the trader or exploiter, then the settler. We do not know what form life may take in other worlds, but we do know that the tragedy of the development of the New World was not the strife between European Nations, but the exploitation, degradation, and in some cases, annihilation of the aborigines.

I could see, in your report, no sign that this danger was appreciated. Rather the objective, perhaps hardly realised, of many speakers seemed to be not the increase of human knowledge, but the desire that man, however changed, however unrecognisable to us, should ultimately dominate more and more of the universe. It is that spirit which finds its slogan in the catchword "The Survival of the Fittest" with the unspoken addition "and by hook or by crook, I'll make sure its me." It is that spirit which C. S. Lewis condemns in his book *Out of the Silent Planet*. It is with that spirit which you, as members of the British Interplanetary Society, must deal.

It may be said that as scientists, you are in no way concerned with the use your discoveries are put to.

That argument would justify the sale of poison to a murderer. Moreover, in publicising their project, your members have taken up an entirely different position. In considering its possibilities, they must consider abuses as well as uses.

It seems certain that in the next fifty years men will be going out to explore space. They will be going to worlds of which it can only be said that whoever is the legal owner, it will not be us. At best, we shall be squatters in deserted premises.

Sooner or later, we shall find inhabited worlds. We cannot say how soon that will be. So it is essential that, before the crisis is upon us, we should be prepared with a policy which will avoid the mistakes we made and the misery we caused in America and Australasia.

T. W. PINNOCK,
Chairman, Society of Model and Experimental Engineers.

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